

COMPARISON OF SIMULATION AND FLIGHT TESTING AT  
AUTOMATIC STOL LANDINGS

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16. Abstract  The comparison of simulation and flight test results for auto- matic STOL landings is presented. The subjects discussed are: (1) description of flight control system, (2) control of flight path during approach, (3) control of aerodynamic flow condi- tions, (4) description of simulator, (5) influence of nonlinear- ity, and (6) application of radar for altitude measurements.			
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# COMPARISON OF SIMULATION AND FLIGHT TESTING AT AUTOMATIC STOL LANDINGS

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## 1. Abstract

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The utilization of extremely short landing strips surrounded by obstacles has become necessary in civil aviation in densely populated areas for reasons of traffic technology and in the military field for tactical reasons. Especially in the battle zone there are often only short landing strips available. The approach to such landing strips must be especially steep on account of lack of freedom from obstacles and guarding against noise. Such steep approaches can only be performed with vertically taking-off and landing aircraft or those having extremely short takeoffs and landings (V/STOL). For the time being there are economical reasons against the use of VTOL aircraft (with the exception of helicopters), whereas STOL aircraft can already be utilized with the promise of success. For this reason, their development is being pushed in many countries.

On account of short runways, one of the most essential requirements for a STOL take-off and landing system is the necessity for being able to hit the touchdown point with great accuracy. With manual landings this precision is largely contingent on weather conditions and becomes increasingly difficult for steep approaches.

In order to fly over obstacles and keep molestation by noise to a minimum, steep, nonstraight line approach profiles become necessary in many cases (Fig. 1). On account of the necessarily

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\* Numbers in margin indicate pagination in the foreign text.

low landing speeds, STOL aircraft must fly with very high lift coefficients closely below the limit of lift for these extreme flight profiles. New navigation and guide-beam systems, as for instance SETAC, make flight navigation along such approach paths possible. In order to fully utilize the flight characteristics of STOL aircraft and the potential of new guide-beam systems, a special flight control system, especially under instrument-flight conditions, is required. Such a flight control system was devised for a proprietary STOL airplane of the DO 28 D Skyservant type by means of comprehensive digital and analog simulation and tried out in a flight test with this plane. In the following, we will show to what extent agreement between test flight and simulation was achieved and what difficulties, especially in the simulation of automatic landing, occurred. /9

## 2. Description of the Flight Control System

Two conditions must be met for the control of STOL aircraft for steep, nonstraight line approach profiles. On the one hand, the deviation of altitude from the desired path and, on the other hand, the aerodynamic flow condition must be controlled.

### 2.1. Control of Path in Approach

During the landing approach, the position of the aircraft can be established by means of elevation and azimuth angle as well as by the distance of the directional radio beacon (Distance-Measure-Equipment, DME) of a guide-beam system or by means of inertial navigation. It is possible to already achieve favorable, non-straight line approach profiles with very simple procedures by means of the SETAC radio guidance system, due to the fact that the flight path is represented in polar coordinates by the angle of beam and the distance of the plane from the directional radio beacon.

For instance, if the rate of acceleration of the controlled angle of beam is constant, the flight path elements at constant flight path velocity consist of parabolas. The vertical accelerations on such flight paths are very small and therefore permit a high degree of passenger comfort. During a guidance-beam approach, the steepest path angles occur, the value of which essentially depend on the drag coefficient of a plane in steep-angle approach and the extent to which it can be controlled, because too small a drag leads to an increase in speed which must be decreased during a long-lasting flattening out procedure. Since during a flight with minimum drag the attainable angles of flight are shallowest, a true steep-angle approach can only be carried out with approach speeds that are less than those for a flight with minimum drag. /10

An adjustment of altitude deviation from the desired path is obtained by means of multiplying the angle deviations by the distance (DME), thereby avoiding the unpleasant bag effect which otherwise occurs during approach on directional radio beam. Flattening out is carried out by means of the radar altimeter signal. The beginning of flare depends on the rate of descent on the glide path, in order to be able to compensate for a constant head or tail wind. Due to the fact that altitude deviations from the desired path during flare must be considerable smaller than during approach, this leads to high speed factors. These high speed factors cause certain difficulties during flight tests, and we will go into details regarding them later on.

## 2.2. Control of the Aerodynamic Flow Conditions

Aside from altitude guidance, an additional problem arises during landings of aerodynamically sustained aircraft during control and influencing of the flow condition at the wing. The plane must be as slow as possible during landing, in order to keep the length of the landing run as short as possible; on the other hand, the fact must be assured that the flow at the wing cannot /11

break away. In a normal case, on the basis of experience and tables, the pilot adjusts the indicated air speed (IAS) as a function of weight, deflection setting, shear wind and turbulence in such a manner that an absolutely safe approach and landing is vouchsafed, while at the same time the length of the available landing run is not exceeded. With conventionally taking-off and landing aircraft (CTOL), the approach velocity is nearly constant and lies approximately in the range of flight with minimum drag.

For STOL approaches, the tolerances of this conventional flight control are too high; on the other hand, an effort must be made to attain a lift coefficient ( $c_A$  value) as high as possible but still safe, and deviations from this  $c_A$  value must be held as small as possible. However, due to the fact that a direct relationship exists between the  $c_A$  value and the angle of incidence, the problem of measuring the  $c_A$  value can be bypassed by determining the proper angle of incidence and holding deviations from this controlled angle of incidence as small as possible.

Controlling the angle of incidence has the advantage over the controlling the velocity in the airstream (speed) in that the pilot no longer needs to consider the effect of weight, flaps, thrust and gusts. Flow conditions at the wing result in the fact that at the same angle of incidence the maximum  $c_A$  value occurs practically independent of deflection setting and blowing of the propeller jet against the wing. For any given case there exists only one optimal angle of incidence for landing and vertical flight for each type of aircraft.

By the feedback of the angle of incidence to the gas throttle of the power plant, the higher frequency gust disturbances cause an unsteady thrust curve which both decreases the service life of /12 the power plant and increases fuel consumption and also disturbs the passengers. In addition, a constantly changing power plant noise is felt by people living adjacent to an airport as being

considerably more annoying than a uniform noise of the same intensity.

The higher frequency gust disturbances can be kept away from the power plant by compensating of the aerodynamic signal by means of a suitable compensating signal (horizontal acceleration).

### 2.3. Wiring Diagram of the Controller

At least two independently manipulated variables are required for the control of two independently controlled conditions -- altitude deviation and angle of incidence; in conventional aircraft these are the elevator and the gas throttle.

In contrast to hitherto existing control systems, divided into stabilizer, automatic pilot and propulsion control, the STOL control system of the Bodenseewerk is strongly coupled and integrated into one unit (Fig. 2). Prior to approach, the barometric altitude deviation, during approach, the deviation from the non-straight line desired path, and during flattening-out, the radar altimeter are being utilized.

Due to the fact that the Skyservant is a tail wheel airplane, it is necessary that touchdown be in the form of a three-point contact with the ground. This can be achieved by adjustment of the angle of incidence by increasing the nominal value of the angle of incidence to the value of the three-point contact position. With a low rate of descent at the touchdown point, the pitch position then corresponds to the angle of incidence.

In accordance with the rule

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$$\gamma = \mu + \alpha - \alpha_w$$

this relation, however, is affected by the gust angle  $\alpha_w$ . However, it is possible to proceed on the assumption that at ground level, vertical air bumpiness is slight.

#### 2.4. Optimization Procedure

The defining of 23 parameters is required for the design of a control system (Fig. 2). This can only be carried out with difficulty when using conventional methods. The evaluation of control systems by means of quadratic cost functions, however, has proved to be practicable. Control systems with minimum values of cost functions (loss functions, criteria of control effectiveness) are defined as optimal. If solved in closed form, the solution of this problem leads to the Riccati differential equation with its known disadvantages for engineering application. These disadvantages can be avoided by use of the following procedure:

For a definitely specified control structure, which has been suitably established on the basis of control-theoretical considerations (e.g. complete feedback of state) and practical viewpoints, the individual and partly modified control areas for the free-to-select control parameters and various quantities characterizing the controlled system are calculated and summed up with weighted factors. The loss functions of the individual kinetic quantities and manipulated variables as well as the parameter sensitivity of the controlled system are evaluated. A numerical search method (minimizing method) varies the control parameters until a minimum of the total control area is obtained. For purely quadratic control areas this iterative solution is identical with the solution of the Riccati differential equation in closed form with very large observation intervals. For this minimum the control parameters with respect to the total area are of necessity insensitive to changes (all partial derivatives are zero).

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This optimizing method basically provides only one optimal, stable setting of control parameters. However, due to the fact that nonsuitable or unnecessary control parameters are technically rendered negligibly small, a structure optimization additionally results by way of elimination of these parameters. Starting from an arbitrarily large structure, qualitative and quantitative comparisons with simpler suboptimal controls are possible. This kind of suboptimal design of controls leads to a very useful compromise from an engineering standpoint.

A control system, and especially an STOL control system, is always a compromise between competing requirements. The requirements for an STOL control system are to the effect that deviations from the command flight profile and the specified angle of incidence be as small as possible and that at the same time adequate quietness of thrust and sufficient passenger comfort are vouchsafed. These competing demands can be formulated by means of criteria of control effectiveness. It is possible to determine control structure and control parameters by means of the automatic optimization method in such a way that the control system meets the demands formulated by the criteria of control effectiveness in the best way possible.

### 3. Description of Simulation

For the purpose of evaluating the control settings arrived at by means of the optimization program, the airplane was simulated on the digital computer in approach position with automatic flattening-out. To this end, DSL 44 (Digital Simulation Language) was used whereby the time slope of all data is presented directly as a result. Due to the fact that in the present problem, especially when on account of the automatic landing the airplane must be simulated over a wide range and from many angles, it is no longer possible to linearize the differential equations. In order to achieve as good agreement as possible with flight

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testing on the one hand but, on the other hand, to keep expenditures for simulation from becoming too great, a set of equations was set up for the airplane which considers the full dependency of dynamic pressure, a parabolic polar curve, the ground effect, dependency of lift on thrust and the dependency of the polar equations on the lift flap setting. These relationships were only partly known prior to the start of simulation and had, therefore, first to be determined from the flight test.

Due to the fact that yawing motion plays a secondary role in the entire STOL problem, the airplane was only represented by longitudinal motion in three degrees of freedom. In flights with high lift coefficients it is also possible to carry out without difficulty a separation of longitudinal and lateral motion.

During simulation of final control elements (throttle lever servomotor, elevator servomotor) it was first attempted to get along with a minimum of effort, that is, they were simulated as lag elements of the first order. The result, however, was that, especially during automatic landing, the final control elements play an important role and must be covered as accurately as possible for simulation (Chapter 4).

The simulation of measuring elements (angle of incidence, rate gyro, accelerometer and guide-beam receiver) proved to be noncritical and the use of lag elements was adequate. One exception, however, was the radar altimeter which is used for automatic landings. Chapter 4 goes into details regarding the difficulties encountered in its use. The guide-beam system was assumed to be ideal in simulation. Practice showed that this was permissible, since it was not possible to determine any effect on the airplane-control system in SETAC.

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Simulating the control itself presented the least difficulties due to the fact that the control could be described mathematically exact without any effort.

#### 4. Comparison of Simulation and Flight Testing

##### 4.1. Stationary Behavior

For the flight test the integrated STOL control was incorporated in the computer installed on board. It was possible to prove that the stationary processes agree well on an approximate basis. It was possible to reproduce well especially the transitional behavior during changes of the ideal value of the angle of incidence and the design altitude. This demonstrates that, on the one hand, the description of the polar curves was adequate and sufficiently exact from the standpoint of input of effort and, on the other hand, that inaccuracies in the description of the airplane are compensated for by the control. At the same time, this is also proof for the fact that an optimized control system, as described, is parametrically insensitive. The strong coupling of controls have shown that changes in the airplane parameter of 30% and changes in the control parameter of 50% will worsen control effectiveness only by about 10%. During flight itself these changes in control parameters can no longer be detected by the passenger in the transitional behavior of the airplane.

##### 4.2. Evaluation of the Criterion of Control Effectiveness

For the purpose of designing the control by means of the automatic optimization method, a criterion of control effectiveness must be defined which evaluates the deviations of the control qualities from the design values, the quietness of thrust and the comfort of passengers. It is an important task of the flight test to prove that a criterion of control effectiveness defined in simulation is sensed in flight as being equally optimal. On this occasion, it transpired that originally passenger comfort in simulation had been evaluated too low. Guide control at the angle of incidence and altitude was so good that the airplane compensated for every gust in pitch position, large angle of pitch velocities occurring in consequence which were felt by the pilot as being very

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disturbing, since he maintains pitch position relatively constant during manual flight. In consequence, a new control was designed for which passenger comfort was evaluated higher. Although this resulted in larger deviations of the angle of incidence and path, the airplane behavior was considerably more pleasant.

#### 4.3. Effects of Nonlinearities

The articulation of final control elements (throttle lever and elevator) entail nonlinearities that consist mainly of slack and friction. They were not taken into account for simulation. During flight testing it became apparent that this neglect is permissible during approach, because then modulation of control parameters is relatively slight. However, for automatic flattening-out, intensification becomes considerably higher during switchover to the radar altimeter in order to be able to maintain great accuracy of touchdown point. On this occasion, we found that the values established in simulation resulted in instability due to the nonlinearities during flight testing. Although reducing the severity of added controls resulted in maintaining a stable system, the required accuracy could no longer be realized. In consequence, it was necessary to eliminate as much as possible the effect of the nonlinearities or to reduce their disturbing influence.

The slack in the throttle lever link amounts to 25% of the /18 total travel. In consequence of this, the original integral wiring of the servomotor resulted in the motor taking a long time (up to 10 sec) for taking up the slack. It was not possible to reduce this slack by structural means. A proportional counter coupling of the servomotor considerably improved the behavior. This resulted in the servomotor, even with small deviations of the control quantities, taking up the slack at maximum running speed. With this configuration the fact must be accepted, however, that the throttle lever oscillates within this slack at a limit cycle.

However, due to the necessary quietness of thrust, the throttle lever is controlled by only low frequency signals so that this entails no negative effects.

On account of an unfavorable design of a chain control, the slack between elevator servomotor and elevator originally amounted to approximately  $4^\circ$ . That is 20% of the total deflection. It was no longer possible to conduct a precisely guided flattening-out. Due to the fact that in the Skyservant the rudder has to overcome large moments from the (electric) servomotor), it was already necessary to proportionally counter-couple the motor in order to reduce its dependency on moments. In connection with the high control factors during the process of flattening-out, this resulted in a limit cycle oscillation of the elevator that made the process of flattening-out impossible. It was possible to reduce this effect to a tolerable quantity by means of careful re-tightening of the chain and reduction of the added control of the radar altimeter by a factor of 3 as compared with the simulation. This demonstrated, however, that the attainable accuracy during automatic STOL landings depends to a great extent on the precision of the linkage of the final control elements.

#### 4.4. Problems Due to the Use of the Radar Altimeter

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The method of flight path guidance at ground level is of decisive importance for automatic landings. Signals from guide-beam system can no longer be received in this range. On account of the required accuracy of  $\pm 1$  m in flight path guidance only very accurately working radar altimeters are suitable. The radar altimeter installed in the Skyservant has an accuracy of 2 ft, which should represent the upper limit for the use for automatic landings. An additional problem is the production of a signal as accurate as possible for the rate of vertical descent, the addition of which during the process of flattening-out is absolutely necessary. The best possibility is to gain it from radar altitude

by differentiation. However, since the radar altimeter used is a digitally operating one, an electric differentiation of the signal varied in the analog range is only possible with great time dealy constants. However, it affects the process of flattening-out to a very unfavorable extent. Here also, measures were required that permitted an automatic landing in the first place and that could not be foreseen by simulation.

In order to derive the exact rate of vertical descent from an electrically differentiated, greatly delayed altitude signal, it is additionally necessary to use a vertical accelerometer, the output signal of which has been delayed by the same time constant.

$$\dot{H} = H \frac{s}{1 + Ts} + K \ddot{H} \frac{1}{1 + Ts} \quad (1)$$

In equation (1) K is a constant, T is the time delay required in order to electrically differentiate H and s is the Laplace operator.

Equation (1) can be transformed into

(2) /20

$$\begin{aligned} \dot{H} &= H \frac{s}{1 + Ts} + H \frac{K \cdot s^2}{1 + Ts} \\ &= H s \frac{1 + K \cdot s}{1 + Ts} \end{aligned} \quad (2)$$

If  $K = T$  an adequately accurate signal for the rate of vertical descent is obtained.

#### 4.5. Comparison of a Flight Trace with Simulation Results

Figs. 3, 4 and 5 show a simulated steep approach with  $\gamma = 6^\circ$  and subsequent automatic landings of a Skyservant. Fig. 6 shows the same on hand of a flight trace plotted in the airplane.

In order to have the possibility of an exact comparison, the flight test would have to be carried out during a complete calm. During the flight test depicted in Fig. 6 a moderate turbulence was obtained and the gusts of wind can be recognized in the angle of incidence and flight.

If the individual data are compared, it becomes evident that altitude  $H$ , angle of position  $\gamma$ , pitch position  $\theta$  and flight  $u$  agree very well. The duration of the flattening-out process is also nearly identical. The angle of incidence is already being somewhat falsified by the gusts of wind, primarily because the angle of incidence sensor exhibits only very weak self-damping characteristics. The center of the curve agrees well with the simulator. As far as the thrust  $S$  is concerned, it is somewhat larger in the flight test at the approach (10%) and, therefore, ~~can~~ must not be increased as much as in the simulation. The reason for the deviation is, on the one hand, due to the accuracy of measurements possible during determination of thrust, and, on the 21 other hand, due to the inaccuracy of the drag coefficient. The elevator curve  $\eta$  does not lend itself to exact comparison, due to the fact that the scale of the test trace is very small. In principle, however, there is no difference.

The progressive increase of  $\eta$  and  $\theta$  after thrust reduction is characteristic for flight test and simulation, in spite of the fact that shortly before, during the flight test, a gust of wind had occurred which is clearly visible in the angle of incidence curve and falsifies it to some extent. After the point of touchdown, quantities can no longer be directly compared, due to the

fact that the landing run cannot be exactly covered in simulation. To this end, the properties of the vehicle would have to be simulated in order to attain agreement.

## 5. Summary

The result of the investigation described here was to the effect that for simulation of the automatic landing the description of the aircraft must be quite comprehensive in order to arrive at comparable findings. In spite of it, automatic landing with the values found in simulation was at first not possible, due to the fact that nonlinearities had a disadvantageous effect. The only solution to this problem was to eliminate as much as possible the interfering influences. It was possible by this means to arrive at a sufficiently far-reaching correlation with the simulation to avoid the necessity of an exact simulation of the nonlinearities. Their effect was only investigated in partial simulations, e.g., in the analog computer.

An automatic landing in accordance with the above-described concept was tried out with the proprietary experimental airplane DO 28 D Skyservant in more than 400 landings to date, and simulation as well as flight testing contributed an essential share to the success of this investigation.

Simulation is absolutely necessary in order to design a control system of this complexity and to define the criteria of control effectiveness. It is the task of flight testing to detect which neglected areas in the simulation are permissible and whether the criteria of control effectiveness have been suitably defined. A close coordination of flight testing and simulation resulted in good agreement in the case of the problem at hand.

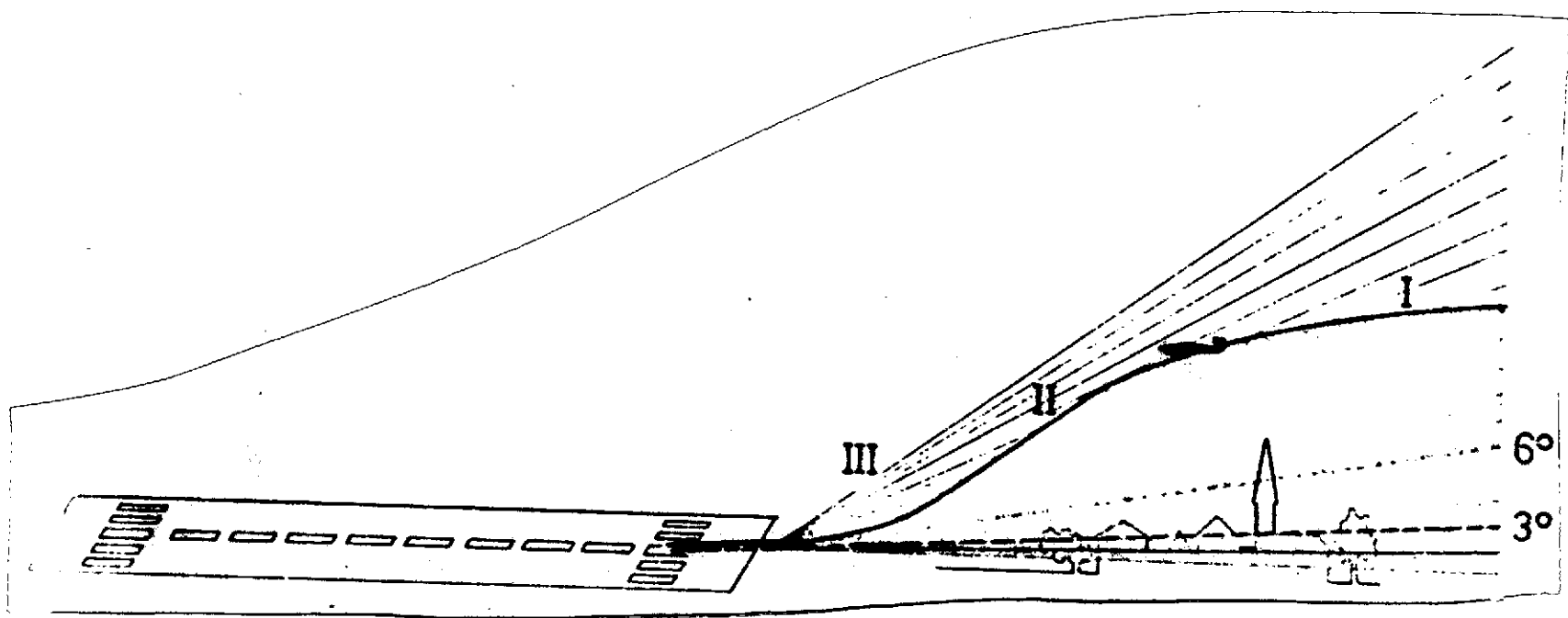


Fig. 1. Nonrectilinear approach profile.

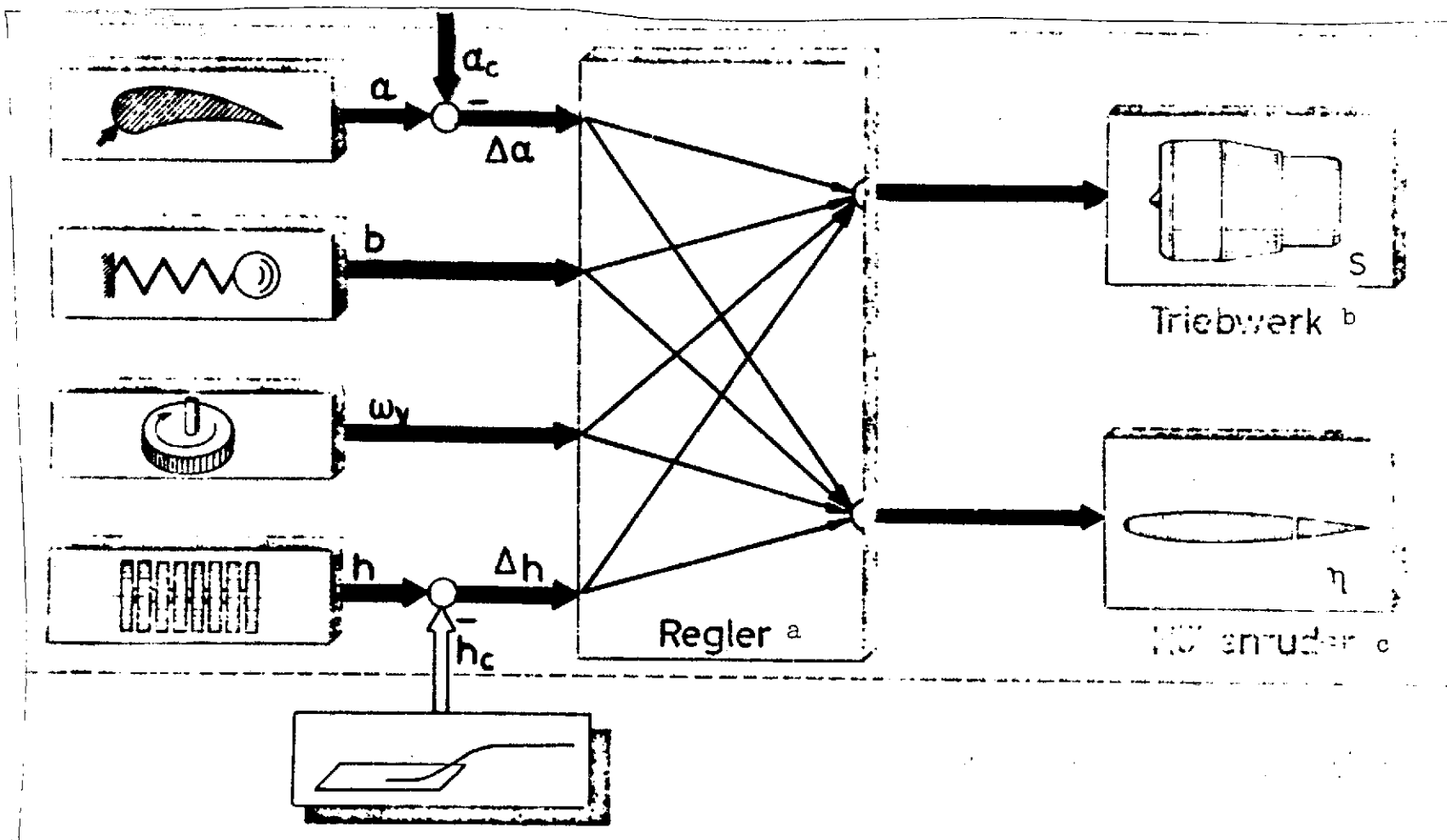


Fig. 2. Integrated STOL flight control system.

Key: a. Control  
b. Power plant  
c. Elevator

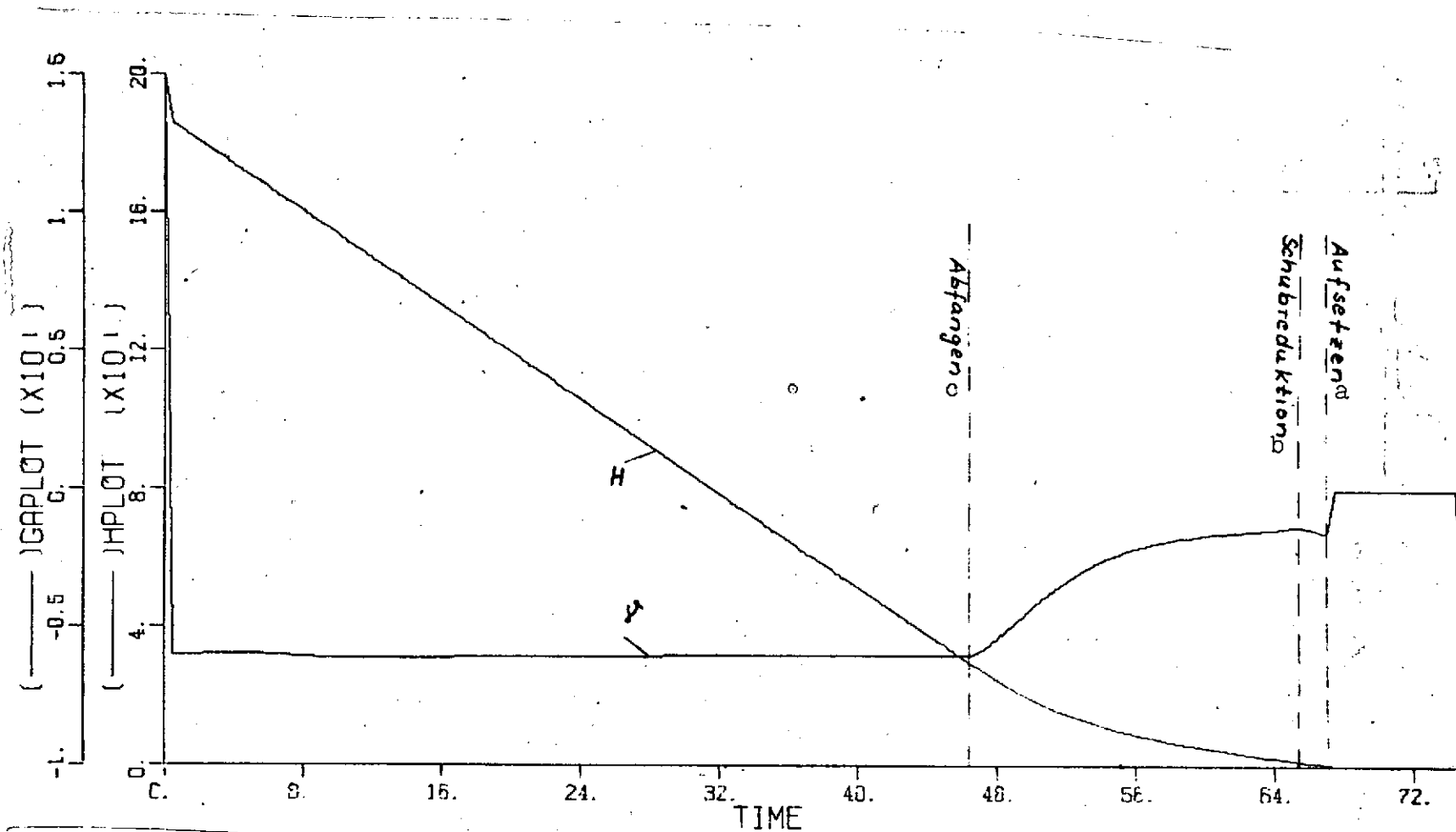


Fig. 3. Automatic landing in simulation.

Key: a. Touchdown  
b. Reduction of thrust  
c. Flattening-out

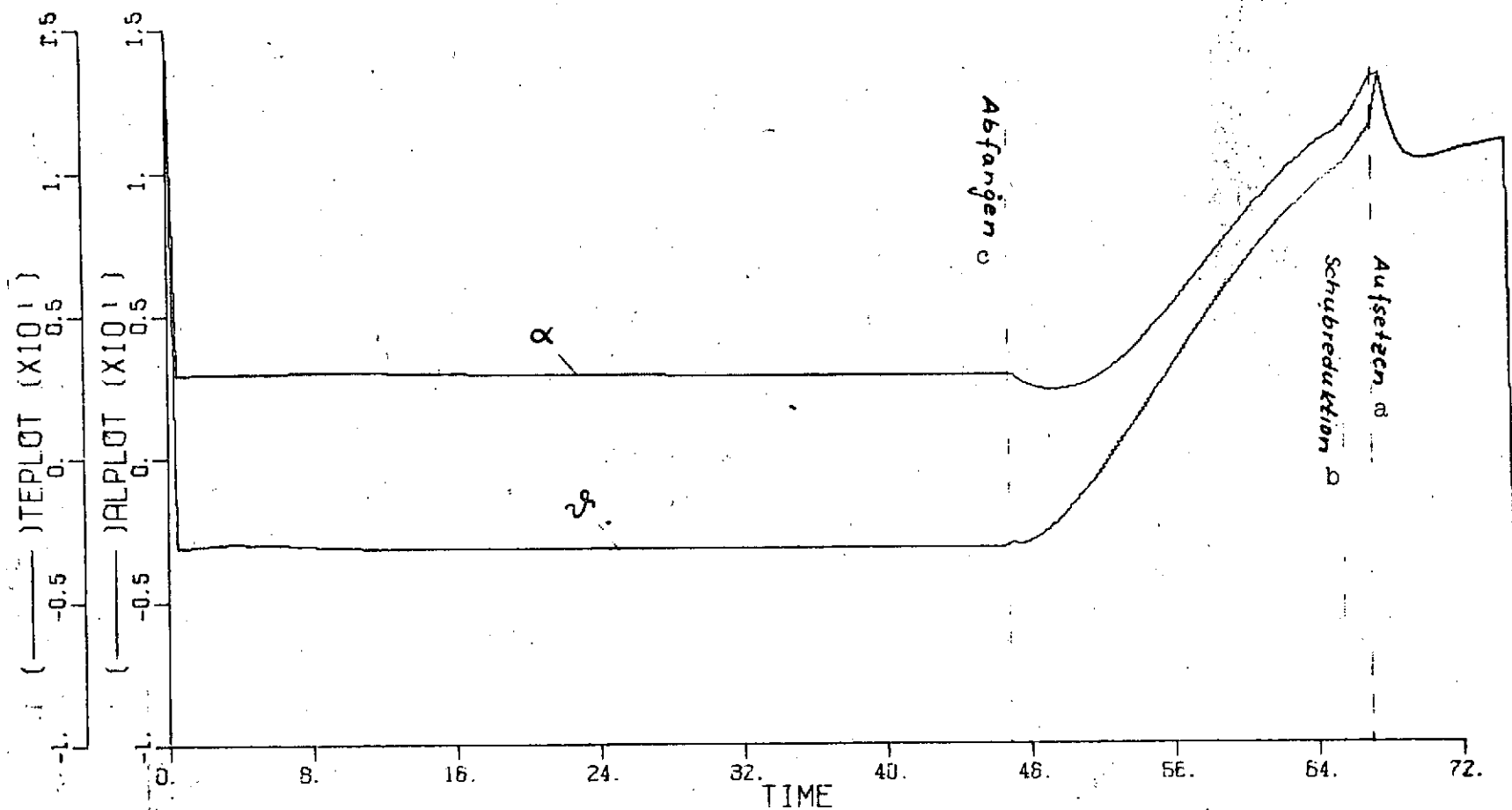


Fig. 4. Automatic landing in simulation.

Key: a. Touchdown  
b. Reduction of thrust  
c. Flattening-out

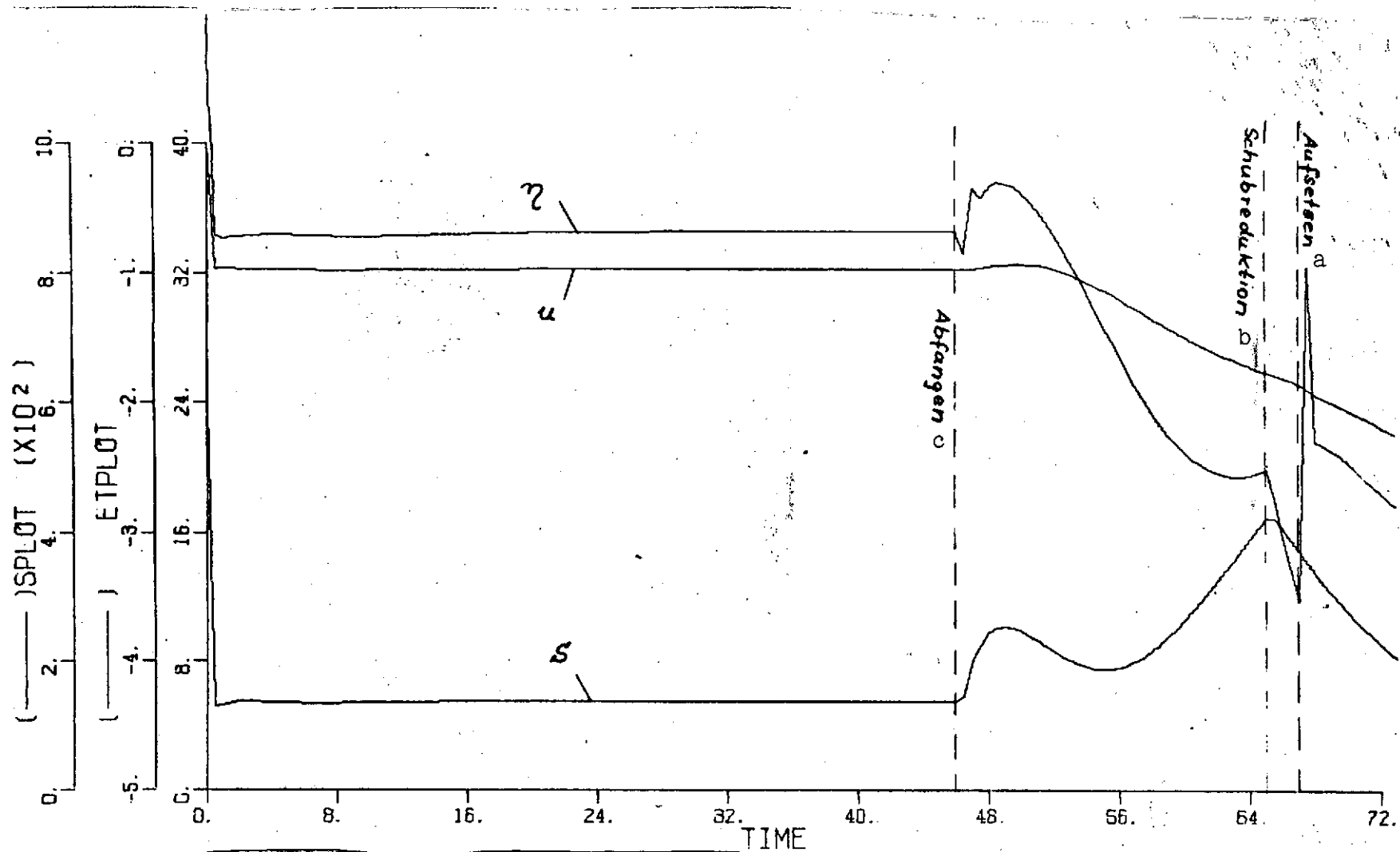


Fig. 5. Automatic landing in simulation

Key: a. Touchdown  
b. Reduction of thrust  
c. Flattening-out

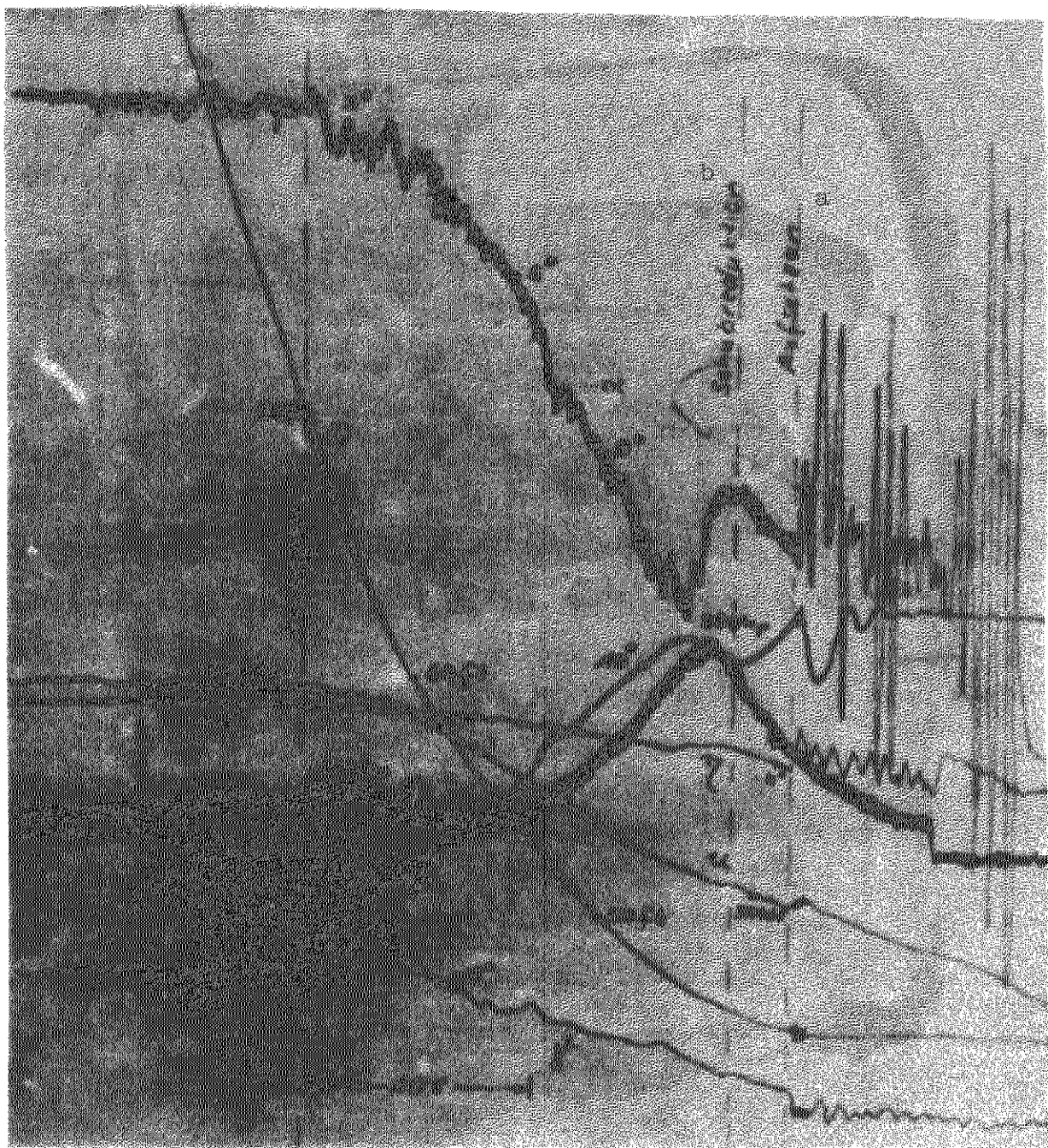


Fig. 6. Automatic landing in flight test.

Key: a. Touchdown  
b. Reduction of thrust  
c. Flattening-out